Correlation of Freestream Pressure Disturbances in Supersonic Wind Tunnels

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Nomenclature

A	= cross-sectional area of nozzle at acoustic origin (see Fig. 2)
A_s	= surface area of nozzle (see Fig. 2)
K_1	= constant in Eq. (4)
K_2	= constant in Eq. (4)
L	= length of nozzle from throat to test section
ℓ_e	= effective length of nozzle (see Fig. 2)
M	= Mach number
n	= constant in Eq. (4)
p	= static pressure
p_c	= local static pressure at surface of cone
r	= radius or height of nozzle
T	= temperature
τ_w	= shear stress at nozzle wall
μ	= Mach angle
θ	= conical half angle of nozzle (see Fig. 2)

Subscripts

a	= value at acoustic origin
aw	= adiabatic wall
c	= conical approximation to nozzle
p	= parallel-conical approximation to nozzle
w	= nozzle wall value
o	= stagnation value
∞	= freestream value
*	= values at $M=1$
(~)	= rms value

Introduction

DISTURBANCE levels in the freestream of wind tunnels at Mach numbers greater than about 2.5 are predominantly pressure disturbances (aerodynamic noise) generated by the turbulent boundary layer on the walls of nozzles.[1] These pressure disturbances have a significant, if not dominant, effect on the values of the Reynolds number where transition occurs on simple bodies.[2-4] A limited amount of transition data taken on cones has been correlated using rms pressure fluctuations measured with a hot wire anemometer in the freestream and pressure transducers mounted flush with the surface of cones underneath laminar boundary layers.[5-7] A generalization of this correlation was suggested in Ref. 6. Unfortunately, there are not enough data available where both transition location and noise levels were measured to completely define the functional dependence of this general correlation on other freestream and local parameters. The suggestion was made in Ref. 6 that if noise data from several different wind tunnels could be correlated, the large mass of transition data available in the literature could be used to better define this functional dependence on other parameters. This Note presents a preliminary correlation of data for freestream disturbance levels in terms of parameters that can be easily calculated for most wind tunnels.

Data Sources

The fluctuating pressure data were obtained from Refs. 1, 6, and 8-11. Data obtained with hot wires in the freestream and with pressure transducers mounted flush with the surface of cones are included. While it is recognized that the boundary layer on the cone at the location of the transducer must be laminar, the proper interpretation of fluctuating pressures measured on the surface of cones is still open to question for several reasons.[6,12-14] Therefore, the hot wire data and surface pressure data will be considered separately.

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Correlation Parameters

The turbulent boundary layers on the walls of supersonic wind tunnels generate pressure fluctuations at the wall and radiate noise into the freestream. The present attempts to correlate the noise data first used theories for calculating the radiated noise from a turbulent boundary layer.[15,16] These theories, with some alterations, were suitable for correlating the low but not the high Mach number data. Therefore, it was postulated that the fluctuating pressures at the wall and in the freestream are related, as in Refs. 7 and 17; and the first step in the derivation of a correlation equation can be expressed as the following identity

$$\frac{\tilde{p}_\infty}{p_\infty} \equiv \frac{\tau_w}{p_\infty} \frac{\tilde{p}_\infty}{\tilde{p}_w} \frac{\tilde{p}_w}{\tau_w} \tag{1}$$

where $\tilde{p}_w/\tau_w$ is based on trends of the data presented in Ref. 18 and on the limiting values at $M_\infty = 0$ and $M_\infty \mapsto \infty$ from Lilly's theory.[19] A relation satisfying the above requirements for adiabatic walls is

$$\frac{\tilde{p}_w}{\tau_{w,\,aw}} = f_1(M_\infty) = 2.2 + 4.1(1 - e^{-0.1M_\infty^2}) \tag{2}$$

The normalized shear stress, τ_w/p_∞, was calculated from turbulent boundary-layer theory[20] using freestream conditions in the test section and distance from the throat of the nozzle to the acoustic origin. The variation of the ratio $\tilde{p}_\infty/\tilde{p}_w$ was obtained from measurements made on the walls and in the freestream of wind tunnels.[18] The low Mach number data for $\tilde{p}_\infty/\tilde{p}_w$ presented in Fig. 1 are from the JPL 20-Inch Tunnel and the total measured values of $(\tilde{p}_\infty)^2$ were "corrected" by dividing by four[22] to represent the noise radiated by one wall of the square test section. The datum point an $M_\infty \simeq 28$ is from an axisymmetric helium tunnel[18] ($M_\infty \simeq 28$ represents the equivalent air Mach number[18] for helium), and since the corresponding "one-wall correction" is not known, the curve was adjusted to a somewhat lower level than the total value shown. The data in Fig. 1 indicate that $\tilde{p}_\infty/\tilde{p}_w$ increases with increasing Mach number, reaches a maximum at some Mach number, and then decreases. This variation of the ratio $\tilde{p}_\infty/\tilde{p}_w$ is significantly different from that used in Ref. 17. However, the trend of the data shown in Fig. 1 seems realistic since $\tilde{p}_\infty$ cannot increase with Mach number indefinitely without resulting in negative instantaneous pressures. On the other hand, $\tilde{p}_w$ can increase significantly without resulting in negative instantaneous pressure at the wall since p_w/p_∞ increases with Mach number.[18] Obviously, a curve fit to the data in Fig. 1 must be somewhat arbitrary.
A third degree polynomial was fitted to the data to give the expression

$$\frac{\tilde{p}_\infty}{\tilde{p}_w} = f_2(M_\infty) = 4.0\times10^{-5}M^3_\infty - 2.478\times10^{-3}M_\infty^2 + 4.125\times10^{-2}M_\infty - 1.234\times10^{-2} \tag{3}$$

which was also required to provide a suitable correlation of the high Mach number data. This equation will undoubtedly require some alteration as more data become available.

The volume of the boundary layer that is radiating noise into the test section and the volume into which the noise is being radiated are important parameters that must be included.

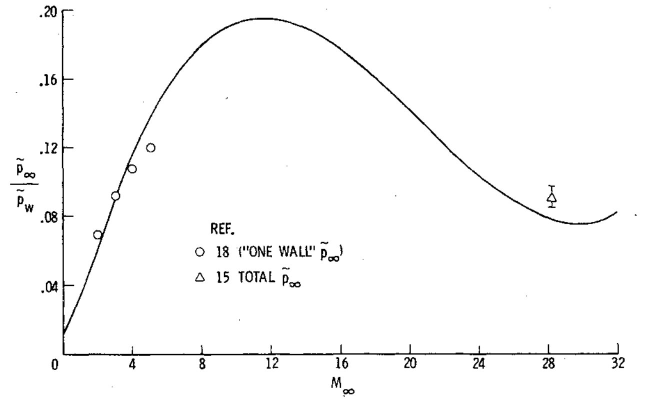

Fig. 1 Ratio of freestream to wall pressure fluctuations measured in wind tunnels as a function of Mach number.

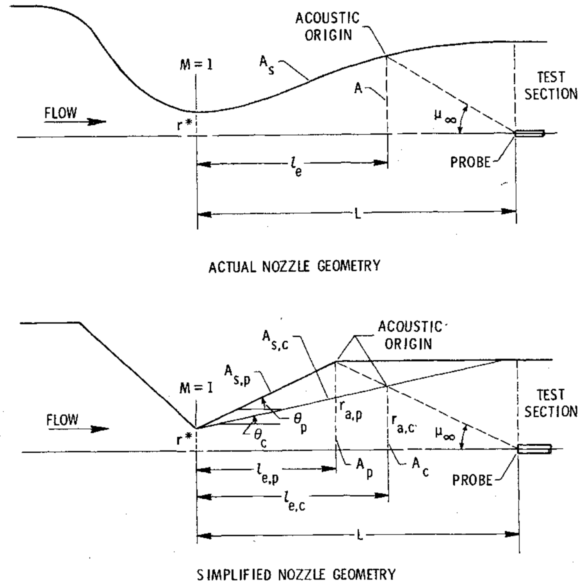

Fig. 2 Approximate geometry for nozzles.

These parameters are characterized by the ratio A_s/A where A_s is the surface area of the nozzle from the throat to the most downstream acoustic origin of disturbances that can be detected by a probe or pressure transducer, and A is the cross-sectional area of the nozzle at that acoustic origin (see Fig. 2). To simplify the specification of A_s/A; the location of the acoustic origin, A_s, and A were obtained by taking the average values of these quantities calculated from conical and parallel approximations to the nozzles as indicated in the lower part of Fig. 2. This averaging process resulted in values for A_s and A that agreed well with the actual quantities for several contoured nozzles.

The equation given by Lilly for $\tilde{p}_w/\tau_w$ includes the effect of wall temperature and fluid temperature evaluated within the boundary layer. In the present correlation this effect is accounted for by the ratio $(T_0/T_w)^n$. The final correlation equation may now be written as

$$\frac{\tilde{p}_\infty}{p_\infty} = K_1 \left[\frac{\tau_w}{p_\infty} \frac{A_s}{A} \left\{ \frac{T_0}{T_w} \right\}^n f_1(M_\infty) f_2(M_\infty) \right] + K_2 \qquad (4)$$

Results

The results of correlating the hot wire anemometer data with Eq. (4) using $n = 0.25$ are shown in Fig. 3. The values of the constants K_1 and K_2 that resulted in the best correlation were 0.0513 and 0.0017, respectively. In this figure and the subsequent one, the total noise level was used. This was done because a suitable method is not known for adjusting data taken in axisymmetric tunnels to the "one wall" case as done by Laufer.[22] The correlation is good at the low noise levels (associated with low Mach numbers), however, at the higher noise levels (higher Mach numbers) there is considerable scatter in the data. Some of this scatter is inherent in the data.[8] The data from Refs. 8 and 11 lie to the left of the line of perfect agreement. This suggests that the correlation could be improved by adjusting Eq. (3) so that values for $\tilde{p}_\infty/\tilde{p}_w$ are larger when the Mach number is about 8. The higher Mach number data taken in helium tunnels scatter around the line of perfect agreement. Thus, while acknowledging these limitations on the high Mach number data, Eq. (4) appears to provide a satisfactory correlation of the hot wire data. Even though some of these data were obtained at $M_\infty = 1.65$, the correlations probably should not be used below Mach numbers of 2.5 or 3.0. This is because, for low Mach numbers, disturbances in the settling chamber are propagated into the test section without adequate attenuation and disturbance

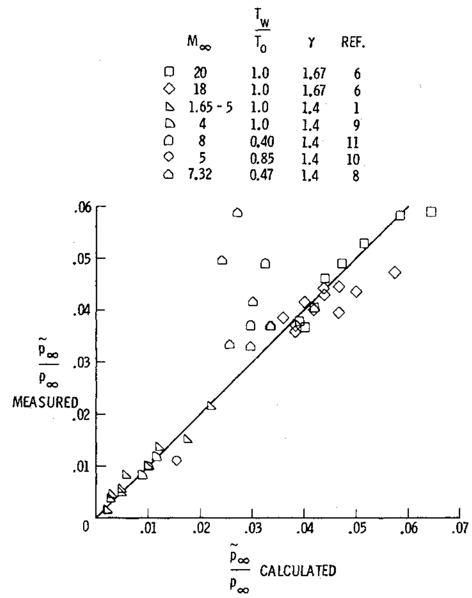

Fig. 3 Correlation of wind tunnel noise data measured with hot wire anemometer.

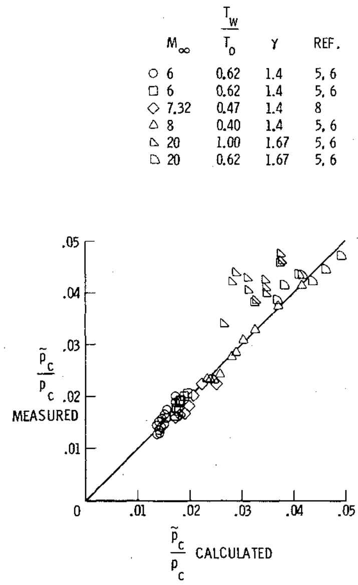

Fig. 4 Correlation of wind tunnel noise data measured with a pressure transducer at the surface of cones.

levels due to other modes could be significant compared to the radiated noise.

The data obtained on cones are presented in Fig. 4. Values for the constants K_1, K_2, and n in Eq. (4) are 0.0355, 0.0033, and 0.25, respectively. The correlation is very good except for the helium data that were obtained at low stagnation temperature. This disagreement between low temperature helium data and the other data is believed to be due to uncertainties in the calculated shear stress used in the correlation.

The present attempt to correlate noise data from supersonic wind tunnels demonstrates the need for more detailed descriptions of facilities in which noise measurements are made. Thus, if accurate values of acoustic origins, flow properties, and geometric factors were available, improvements in the correlation should be possible. Meanwhile, the correlations of wind tunnel disturbances presented herein will provide useful predictions of rms pressure disturbances in the freestream of wind tunnels or on the surface of cones at Mach numbers greater than 2.5.

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